TECHNICAL NOTE

D-1018

A NOTE ON HELICOPTER ROTOR-BLADE

FATIGUE-CRACK PROPAGATION RATES UNDER

EQUIVALENT-LIFETIME FATIGUE LOADINGS

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON   February 1962

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SUMMARY

This paper presents the results of a brief investigation of the rela-
tive rates of fatigue-crack propagation obtained in helicopter-rotor-blade
fatigue tests in which simplified, equivalent-total-lifetime, fatigue-test
loadings at zero mean load are used to simulate a flight fatigue loading
that includes a mean tension load. The results indicated that the conven-
tional equivalent-lifetime loadings do not give equivalent rates of crack
propagation. For typical rotor-blade loadings, which included large mean
tension load, the general trend was toward greatly reduced rates of
fatigue-crack propagation under the equivalent-lifetime loadings. The
crack propagation rates obtained under the conventional equivalent-
lifetime loadings provided a nonconservative basis for establishing
rotor-blade inspection intervals.

INTRODUCTION

Helicopter-rotor-blade structural loadings for any given flight con-
dition consist of oscillatory loads superimposed upon a steady mean load.
The oscillatory loadings are periodic in nature and predominate at fre-
quencies that are harmonics of rotor rotational speed. The mean tension
load is associated with the steady aerodynamic and centrifugal loadings
on the blade.

In conducting the fatigue substantiation of rotor blades, it is the
general practice to apply a combination of oscillatory loading and mean
tension loading in order to simulate a critical flight loading in the
laboratory fatigue test. The lifetime of concern in these tests is the
number of cycles of applied loading sustained until blade rupture. This
total lifetime includes initiation and propagation of a fatigue crack in
the blade specimen. In the event that application of the combined oscil-
latory loading and mean loading is not feasible, a widely accepted alter-
native has been to apply an equivalent oscillatory loading without the

mean tension load. The equivalent oscillatory loading is selected on the basis that it will give the same total lifetime to specimen rupture as will the combined oscillatory and mean tension loadings.

In the past, the principal efforts in rotor-blade fatigue substantiation have been in the direction of establishing a safe service life based upon lifetime to blade rupture. Recently, added emphasis has been given to the possibility of increasing service reliability of rotor blades by frequently inspecting the blades to detect fatigue cracks prior to catastrophic failure. The selection of the safe inspection interval is based upon the rate at which the fatigue crack propagates through the specimens under flight loading conditions.

When establishing inspection intervals, the question arises as to the reliability of the fatigue-crack-propagation information obtained with the use of the conventional equivalent-lifetime loadings. A brief investigation was undertaken in order to determine the trend in the relationship between crack propagation rates under a typical flight loading and the rates achieved under conventional equivalent-lifetime loading. Two crack propagation tests were conducted on specimens cut from a production rotor-blade spar. The results of the limited tests are compared with existing semiempirical methods for calculating crack propagation rates in aluminum alloys presented in references 1 and 2.

SYMBOLS

N number of cycles of load application

R ratio of minimum nominal stress to maximum nominal stress in fatigue load cycle, S_{min}/S_{max}

r_{flight} crack propagation rate under simulated flight loading, in./cycle

r_{test} crack propagation rate under equivalent-lifetime loading, in./cycle

S_a amplitude of nominal alternating stress, ksi

S_m mean nominal stress in fatigue load cycle, ksi

S_{max} maximum nominal stress in fatigue load cycle, ksi

S_{min} minimum nominal stress in fatigue load cycle, ksi

S_u ultimate tensile stress, ksi

S_y tensile yield stress, ksi

Subscripts:

1   flight loading

2   equivalent-lifetime loading at $R = -1$

SPECIMENS AND PROCEDURE

The specimens used in the fatigue-crack propagation tests were cut from the upper and lower surfaces of a new extruded rotor-blade spar section as indicated in figure 1. The spar section was of extruded 6061-T6 aluminum-alloy material. A stress concentration was introduced by drilling a 1/8-inch-diameter hole in the center of each specimen. The fatigue-crack propagation tests were conducted with a subresonant axial-load fatigue machine which had a cyclic loading rate of 1,800 cycles per minute.

One specimen was tested with a loading which was representative of maximum oscillatory loading and associated mean tension loading encountered in normal helicopter flight operations. This simulated flight loading corresponded to a mean nominal stress of 8 ksi and an alternating stress of ±5 ksi in the specimen. The equivalent-lifetime loading for the second specimen was selected in the conventional manner from a constant-lifetime diagram similar to that shown in figure 2. In actual practice, the constant-lifetime diagram used must be for the same material, specimen size, and stress-concentration factor as those existing at the critical section of the rotor blade. In selecting equivalent-lifetime loadings from a diagram of this type, a total lifetime to specimen rupture is determined for conditions of alternating-stress level $S_{a,1}$ and mean tension stress $S_{m,1}$ which constitute the flight loading condition. Moving along the constant-lifetime curve, indicated by the series of arrows in figure 2, to the axis $S_m = 0$ establishes the new or equivalent alternating-stress level $S_{a,2}$ to be used in the laboratory fatigue test. The value of $S_{a,2}$ which was determined in this manner and used as the nominal stress for the second specimen tested was ±8 ksi.

Each of the two specimens was loaded cyclically until fatigue cracks became visible at the edges of the central hole. Loading was continued and crack propagation was carefully monitored with the use of 30-power magnification and stroboscopic lighting. In order to have positive identification of the cracks and to avoid localized effects near the

edge of the hole, a total crack length of 0.2 inch (including the central hole) was chosen as the initial crack length for monitoring purposes. Monitoring was then continued until the rate of fatigue-crack propagation indicated imminent rupture of the specimen.

RESULTS AND DISCUSSION

The results of the crack propagation tests are shown in figure 3. In this figure the percent of cross-sectional area lost as a result of fatigue-crack propagation is plotted against the number of load cycles applied after the crack had reached a length of 0.2 inch.

The number of load cycles between a crack length of 0.2 inch and total failure for the equivalent-lifetime test loading was approximately eight times that for the simulated flight loading. If a rate of cyclic load application in service of once per rotor revolution is assumed, the crack propagation curve for the equivalent-lifetime fatigue test loading represents approximately 150 hours of flight time as a basis for establishing a safe inspection interval. However, under the simulated flight loading with mean tension load acting, the time from initiation of a visible crack to failure of the specimen was only 20 hours. Thus, the technique of using the constant-lifetime diagram for determining equivalent-lifetime fatigue loadings resulted in a significant reduction in the rate of crack propagation and therefore provided a nonconservative basis for establishing blade inspection intervals.

The results of the limited crack propagation tests were compared with the more general results of a fatigue-crack propagation investigation presented in references 1 and 2. The semiempirical equations of references 1 and 2 for calculating crack propagation rates were considered to apply inasmuch as they were established for values of R or $\left(\dfrac{S_{min}}{S_{max}}\right)$ of 0 and -1. The data for $R = -1$ (completely reversed loading) actually duplicate the equivalent-lifetime loading condition at zero mean load. The reference data for $R = 0$ adequately approximate the simulated flight loading condition for the purpose of demonstrating trends in relative rates of crack propagation.

The ratios between rates of crack propagation under the simulated flight loading and the rates under the equivalent-lifetime loading were calculated by the semiempirical equations presented in references 1 and 2. The calculated ratios for 2024-T3 and 7075-T6 aluminum-alloy specimens are presented in figure 4 as a function of the specimen cross-sectional area lost because of crack propagation. The results of the actual 6061-T6 blade-specimen tests are also presented in figure 4.

In general, the results indicate that the crack propagation rates in flight are many times greater than the rates obtained in equivalent-lifetime tests at $R = -1$. During the early stages of crack propagation, the semiempirical calculations indicate that for 2024-T3 aluminum alloy the crack propagation rate in flight is approximately four times the rate obtained in the equivalent-lifetime test. For the 7075-T6 material, the semiempirical calculations indicate a rate of crack propagation in flight approximately six times the rate for the equivalent-lifetime test, and for the 6061-T6 aluminum-alloy blade-specimen tests, the crack propagation rates for the flight loading were as high as 10 times the rate for the equivalent-lifetime fatigue test.

The results of the semiempirical methods of references 1 and 2 and the two fatigue-crack propagation tests indicate that, in general, the conventional method of selecting test loadings at $R = -1$ to simulate flight loadings with positive mean load will result in reduced rates of crack propagation. For the case illustrated in this paper, the rates of crack propagation and the associated number of cycles from crack initiation to catastrophic failure under the conventional equivalent-lifetime loading gave highly nonconservative results upon which to base the selection of safe rotor-blade inspection intervals. The reduced rates of crack propagation point to the need for caution when interpreting the results of simplified fatigue tests.

As pointed out in reference 2, for 7075-T6 aluminum alloy it is the simulation of the tension portion of the fatigue loading cycle that is the major factor in achieving equal crack propagation rates for values of load ratio $R = 0$ and $R = -1$. Therefore, if it is not feasible to duplicate a flight loading condition at $R = 0$ in the laboratory by including the mean tension load, an adequate increase should be made in the amplitude of the oscillatory loading at $R = -1$ during the crack propagation phase of the tests to achieve truly equivalent rates of crack propagation. The increase in amplitude of the loading at $R = -1$ would be such that the maximum stress would be equal to the maximum stress in the flight loading at $R = 0$. As also noted in reference 2, for 2024-T3 aluminum alloy the situation is generally the same, but the maximum stress at $R = -1$ had to be reduced below the maximum stress at $R = 0$ in order to achieve equal rates of crack propagation. Further investigation will be required to determine the specific values of loading at $R = -1$ that are necessary to achieve crack propagation rates equal to rates obtained at other values of R and with other materials.

CONCLUDING REMARKS

The results of a limited investigation of rotor-blade crack propagation indicated that the conventional equivalent-lifetime fatigue loading

at zero mean load, which was selected on the basis of lifetime to total failure, did not give crack propagation rates equal to those which would be experienced under the given flight loading with a steady tension load acting. The tests indicated that the time between a crack length of 0.2 inch and total failure for the equivalent-lifetime loading was about eight times that found for the simulated flight loading. Use of this type of equivalent-lifetime loading procedure for laboratory tests thus provides a nonconservative basis for estimating rotor-blade inspection intervals.

Langley Research Center,
　　　National Aeronautics and Space Administration,
　　　　　Langley Air Force Base, Va., November 14, 1961.

REFERENCES

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2. Illg, Walter, and McEvily, Arthur J., Jr.: The Rate of Fatigue-Crack Propagation for Two Aluminum Alloys Under Completely Reversed Loading. NASA TN D-52, 1959.
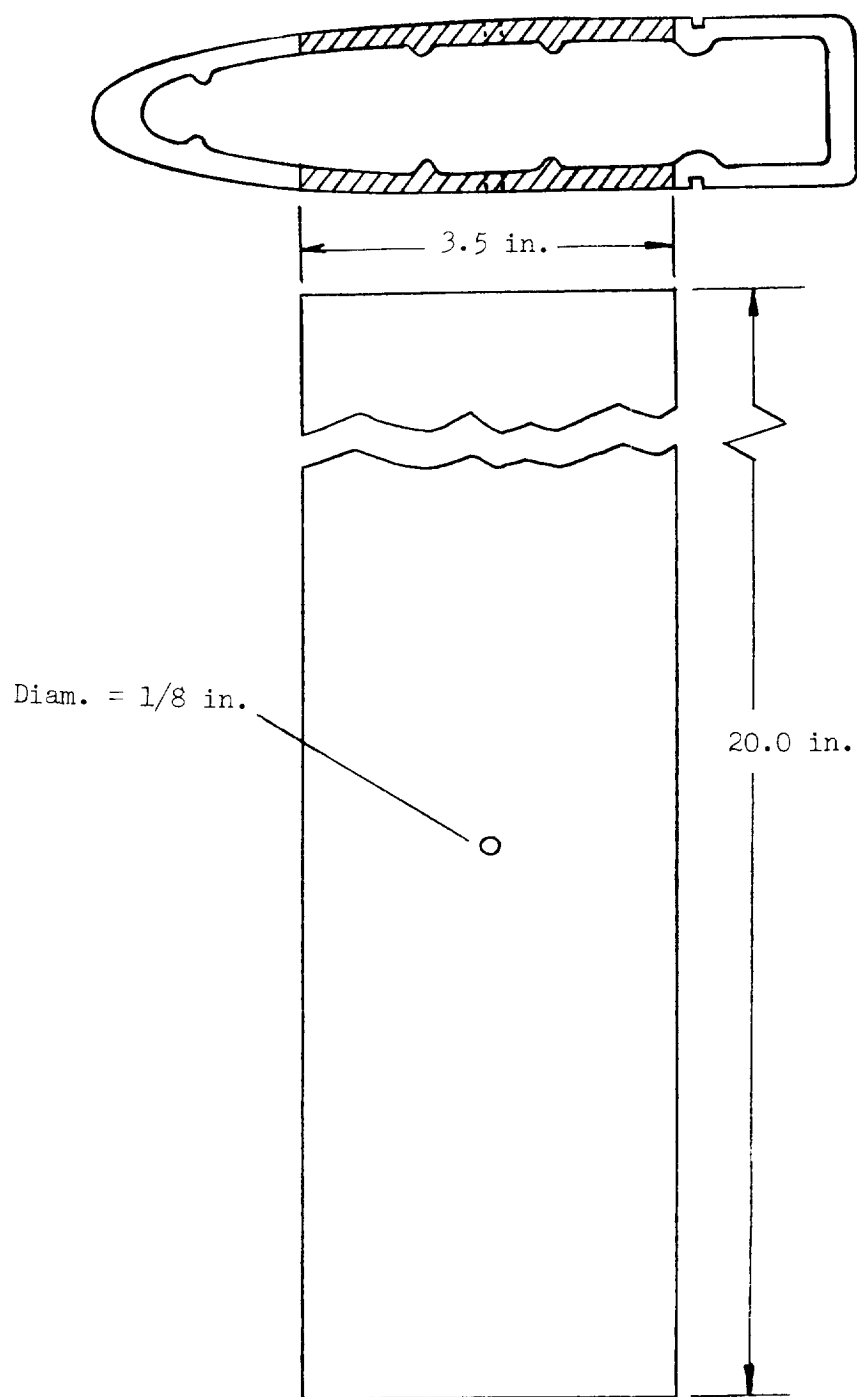
Diam. = 1/8 in.

3.5 in.

20.0 in.

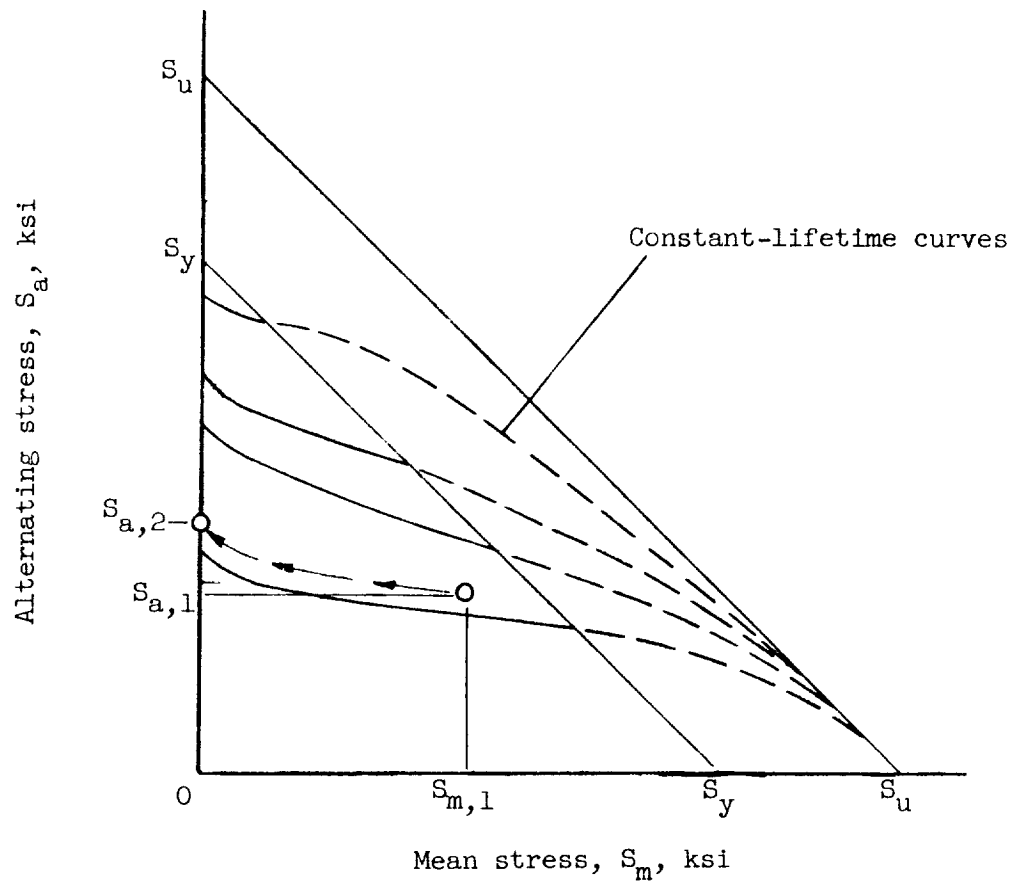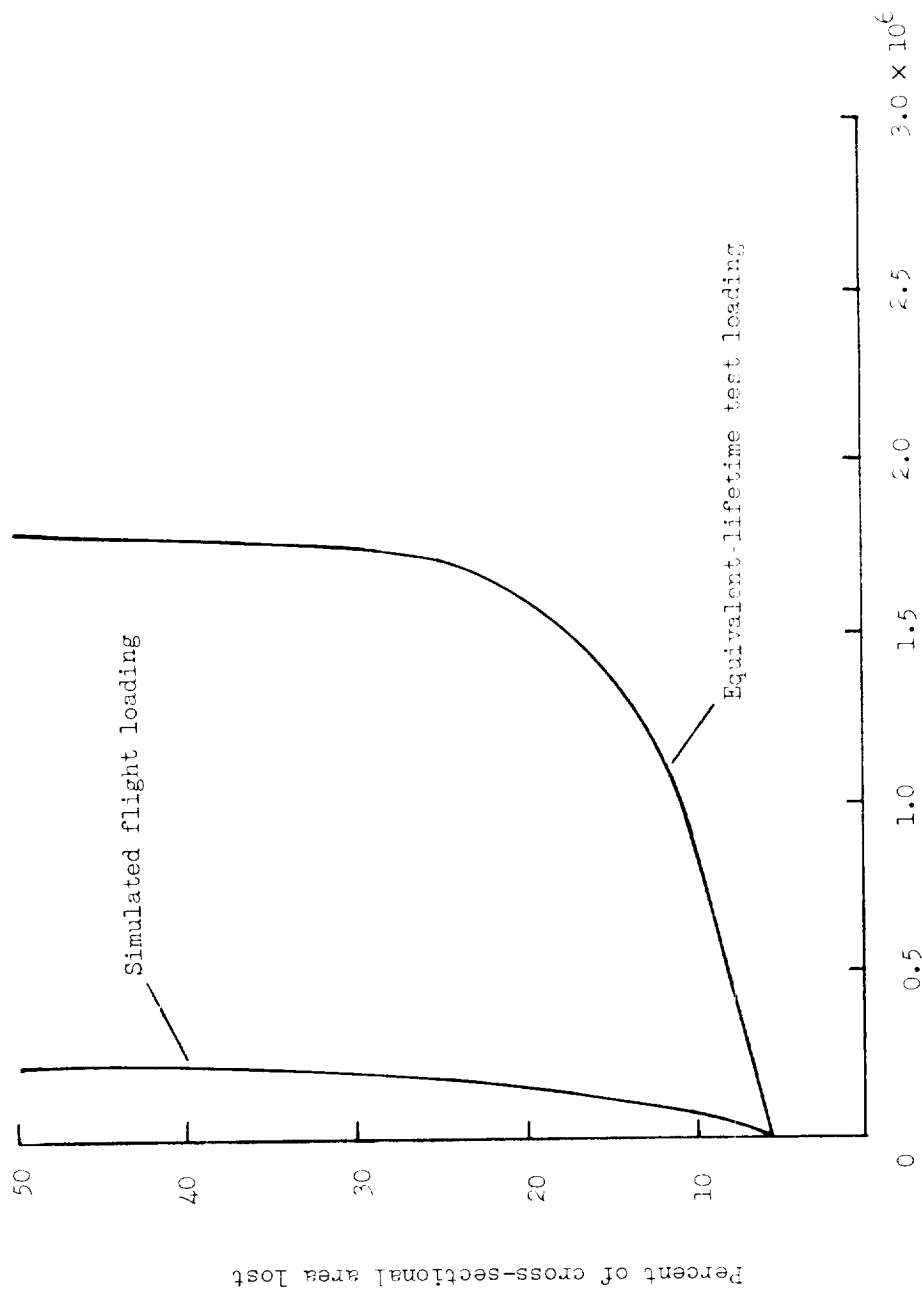Figure 1.- Fatigue-crack propagation specimens.

Figure 2.- Constant-lifetime diagram.

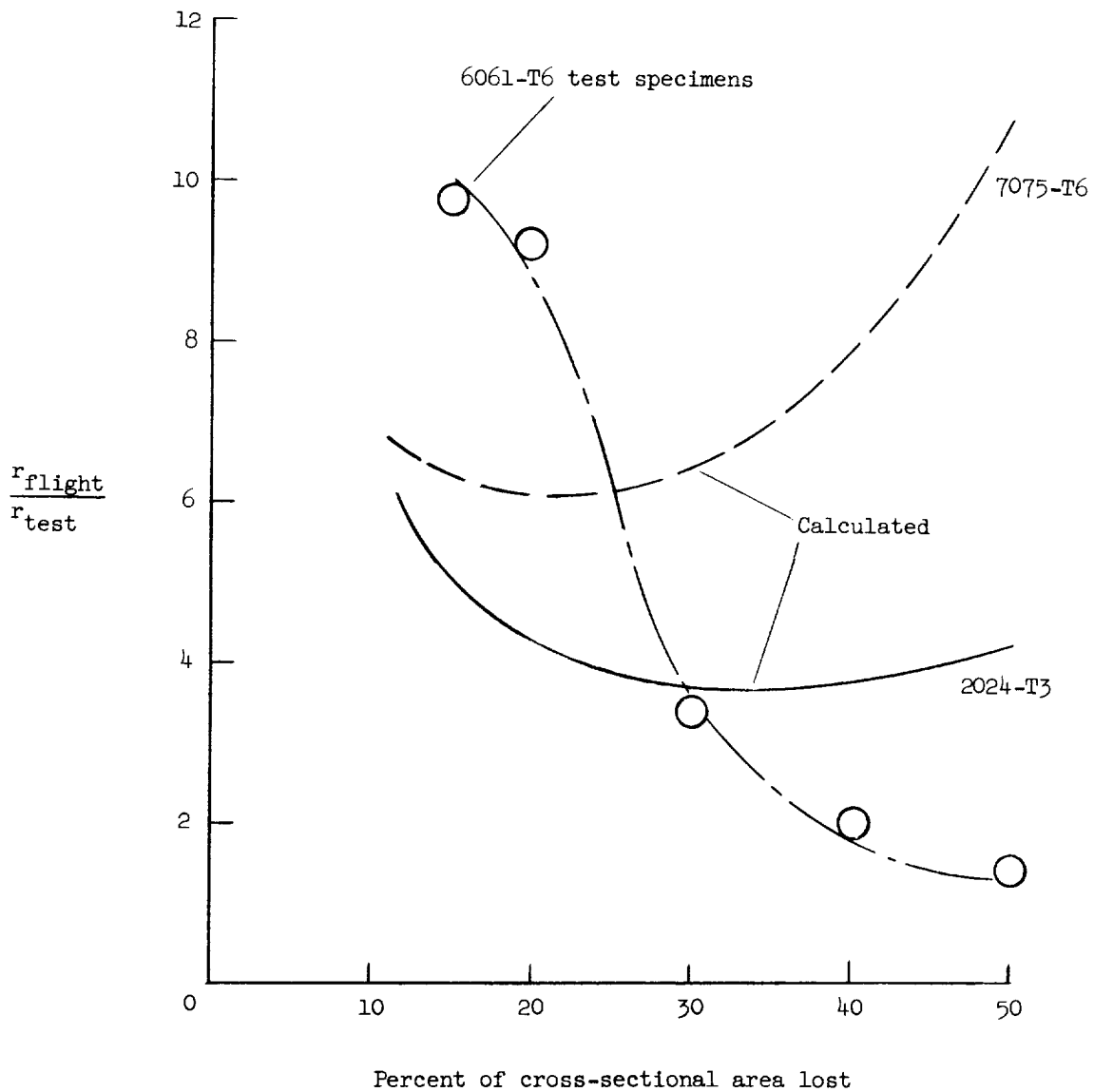N, cycles after 0.2-inch crack length

Figure 3.- Crack propagation curves.

Figure 4.- Ratio of the rates of crack propagation in flight to the rates in an equivalent-lifetime fatigue test. Calculated rates were obtained from references 1 and 2.